STABILITY

WITH

COMBUSTION AND FLOW

by

Lieut. Comdr. John B. Balch, U.S.Navy

June 1, 1948

Thesis
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Table of Contents

Stability with Combustion and Flow

Abstract

This report contains the results of an experiment with a given burner to determine some of the factors which effect combustion stability, combustion efficiency, and blow out. Investigations were made on the effects of mass flow, burner pressure, chamber length, and fuel pressure.

It was found that burner pressure, shorter burner length, smaller weight flows, and more volatile fuels improve blow out conditions. It is believed that extra oxygen supplied to the burner when using a hydrocarbon fuel will also improve this limit. On the other hand, longer burner length is necessary for improved combustion efficiency. In this particular burner high weight flows, and high air-fuel ratios badly reduced efficiency, due to the improper introduction of the primary air.

The investigation was conducted by Lieutenant Commander J. B. Balch at the Mechanical Engineering Department of Rensselaer Polytechnic Institute, Troy, New York. The necessary background information in the thermodynamics of high velocity flow and combustion was received

from Professor Neil P. Bailey, the head of the Mechanical Engineering Department.

Introduction

The development of the ram-jet, turbo jet, and to some extent the liquid fuel rocket during World War Two, brought to the fore the problem of combustion with high velocity flow. The first glance at the problem would lead one to believe, that the principles of static combustion which have been worked out so thoroughly in the past, could be applied directly to the flow combustion problem. Unfortunately this is not so.

In the first place there are certain flow conditions for any burner that must be met. These include the unavoidable static pressure drop in a combustible chamber, the partial velocity pressure gain for that static pressure drop, friction pressure drop, the degree of turbulance due to a burner design, the choking limit of temperature rise, and the velocity distribution in the burner. The action of the flame in the boundary layer is another subject associated with flow and not found in the static combustion problem.

Secondly, the chemical processes of combustion are altered in that the time element of the combustion process has been shortened and therefore the reaction may be only partially completed in the combustion chamber. As will be seen later this has a serious effect on blow-out. Added to this are the problems of burning a heterogeneous

mixture at velocities greater than the transformation velocity.

The problem of the jet engine burner is to allow a fuel to burn quickly and completely in fast moving air over as wide a range of conditions as possible.

The burner used in these tests was an experimental 2in. General Electric combustion chamber as shown in Fig. 1. Kerosene was used as the fuel. The purpose of the experiment, as originally laid down, was to attempt to correlate instability and blow-out with flow theories or to correlate it with the limitations in the mechanics of combustion. However, as will be explained later, the instability in this particular burner was found to be caused by the mechanics of combustion only. As a result the experiment was broadened along these lines to include burner efficiencies at various operating points.

Instability as used in this report refers to the point where the burner is on the verge of blowing out. In other words, it does not refer to slightly rough burning.

Equipment and Procedure

The burner was a typical gas turbine type as shown in Fig. 3. The shell was a 2 in. inside diameter steel cylinder built in two sections. The forward or up-stream section contained the liner, fuel atomizer and spark plug and attached to this was the after or down stream section, 6 $\frac{1}{4}$ in. long. This 6 $\frac{1}{4}$ in. part of the burner was added to insure that the combustion reaction had gone to completion for the arbitrary air-fuel ratio of 50 to one with a .6 in. diameter exit nozzle.

Various exit orifices of .4 in., .6 in., and .7 in. in diameter were used at the end of the down stream section. Static pressure taps were placed at the end of the forward and after sections, while temperature taps were placed at the end of the forward section and the beginning and the end of the after section. A three dimensional stand was placed at the end of the burner to read total and static pressure and total temperature at the orifice, inside the burner or in the free jet.

The stainless steel liner fitted into the forward shell and was held in place by the nozzle spray extension which screwed into the back of the liner. The liner had eight rows of sixteen holes, as shown, through which the air diffused into the combustion chamber. The holes

increased in diameter from 1/8 in. at the forward end to 1/4 in. at the after end. Through the side of the burner and liner and just down stream of the fuel spray a spark plug was placed.*

Connected to the forward end of the burner by a 1 in. pipe was the plenum chamber with a total temperature Weston thermometer and a .749 in. diameter metering nozzle. Also connected were the necessary static pressure taps to compute weight flow. Ahead of this chamber were the air control and bleed off valves.

Associated equipment was a fuel tank mounted on scales with a vacuum fuel feed line, fuel pump, motor, variac, spark coil, potentiometer, selector switch, and manometers.

The air supply was delivered by a Schram Compressor at 100 lbs. per sq. in. and 205 c.f.m. During the runs the compressor was kept running at a constant delivery pressure by bleeding off excess air.

The following procedure was used. The spark was turned on followed by fuel and air. The fuel was adjusted to the desired delivery pressure and then the air supply to the metering chamber was adjusted to give the

* The burner liner shown in Fig. 3 had been badly burned and broken by blow out tests.

desired manometer reading. With this done the air bleed off valve was opened to give a steady 90 lb. delivery air pressure. Final adjustments of fuel and air were then made. When conditions were steady, the readings were taken.

Fuel weight was obtained by the scales in the following manner. When the arm of the scales passed a scribe mark a two ounce weight was removed and a stop watch started. As the pointer arm again passed the scribe mark the watch was stopped. Thus, knowing the time for the consumption of two ounces of fuel, the lbs. of fuel per second could be computed.

The intake fuel line and the by pass fuel pump line were suspended in the fuel tank. For the amount of fuel used the error introduced by this arrangement was negligible, being a function of the density of the copper pipe and kerosene, and the difference in momentum of the entering and departing fuel. The system was calibrated by direct measurement of fuel delivered.

For the few hydrogen and propane runs the procedure was the same except that the gas was not metered because time was insufficient to construct an adequate fuel metering system and to calibrate it.

Occasionally, when computing data, it was found necessary to correct the weight of fuel used per sec.

This was accomplished by a plot of fuel versus burner static pressure for constant fuel pressure runs. The point obviously in error was moved to the approximately straight line relationship between burner pressure and weight of fuel. If more than one reading was out, the run was not used.

All burner temperatures T_2 are the maximum temperatures in the burner and are uncorrected.

On each regular run the following data was taken. The static pressure ahead of the metering nozzle was recorded on a 60 in. mercury manometer. The changes in static pressure across the metering nozzle, from the discharge side of the nozzle to a point half way along the burner, and to the end of the burner tube were recorded by differential manometers. Total temperatures were taken at the half way point in the burner, at the end of the burner, and in the plenum chamber on the discharge side of the nozzle. Fuel weight was taken as previously described.

Many burner temperature traverses were taken in attempts to correlate an average temperature in the burner with air and fuel flow conditions. They were found impractical, as,for a true average temperature, it would have been necessary to compute a double integration for temperatures and velocity across the burner

for each flow condition. However, these traverses were useful in determining the position for the temperature probe to read the maximum temperature at any flow condition. See Fig. 11. In evaluating results then it must be remembered that the temperature at the end of the burner is the maximum and not a weighted average, which would be from 200 to 500 degrees less than the maximum.

For the same reason no orstat analyses are included as measures of combustion efficiencies. However, by plots shown relative efficiencies are determined in which point this report is interested. It can be stated, however, that a traverse across the burner indicates that combustion efficiency is highest at the center, low at the top, and lower still at the bottom where in some cases liquid fuel was actually seen.

Discussion and Results

Although, as previously pointed out, instability in this burner was eventually found to be connected with the kinetics of combustion, a short review of a few of the theoretical flow limitations is in order.

In reference (a) and (b) it is shown that for frictionless constant area burning

$$\frac{P_1-P_2}{2} = \frac{\rho_2 V_2^{\,2}}{2} - \frac{\rho_1 V^2}{2}$$

which states that the gain in velocity pressure during burning is equal to half the static pressure loss. At high velocity burning therefore the total pressure loss will be severe. It is further shown that

$$\frac{dM}{dTo} = \frac{M\left[1+\left(\frac{k-1}{2}\right)M^2\right]\left[1+kM^2\right]}{2\ To\ \left[1-M^2\right]}$$

for the above conditions, which indicates that the subacoustic mach number increases with increase in total temperature until M=1, which point it approaches at an infinite rate.

It is shown in reference (b) that $\frac{W}{P}\ \frac{dP}{dW}$ must be negative in a burner for stability. If one differentiates the weight flow equation $\frac{W\sqrt{To}}{AP} = M\sqrt{\frac{kg}{R}}\left[1-\left(\frac{k-1}{2}\right)M^2\right]$ and combine it with $\left(\frac{Po}{P}\right)^{\frac{k-1}{k}} = \left[1+\left(\frac{k-1}{2}\right)M^2\right]$ differentiated for constant total pressure one finds that $\frac{W}{P}\ \frac{dP}{dW} = -\frac{1}{k}\left(1-\frac{1}{M^2}\right)$ This states that if one burns at a mach number greater than one, the flame will be unstable.

In reference (d) it is shown that

$$\left(\frac{R_2}{R_1}\right)^{\frac{1}{2}}\left(\frac{To_2}{To_1}\right)^{\frac{1}{2}}\frac{\left[1+\left(\frac{k-1}{2}\right)M_1^{\,2}\right]^{\frac{1}{2}}M_1k_1^{\frac{1}{2}}}{(1+k_1M_1^{\,2})}=\frac{\left[1+\left(\frac{k-1}{2}\right)M_2^{\,2}\right]^{\frac{1}{2}}M_2k_2^{\frac{1}{2}}}{(1+k_2M_2^{\,2})}$$

This equation gives the maximum M for any $\frac{To_2}{To_1}$ where $M_2 = 1$. It can be considered a limit of stability above which choking will occur. The superacoustic solution to the foregoing equation has previously been eliminated.

As runs were made it became obvious, that in this particular burner, that instability was not the result of the violation of thermodynamic flow requirements. One of the strongest proofs of the above observation was that at rich blow out the flame was extinguished progressively from the down stream end. Propane gas was then used instead of kerosene with identical results. However, when hydrogen was burned greatly increased burner temperatures were attained and the flame at blowout was extinguished from the up stream end. It is believed that this later fuel was giving flow instability.

Typical data taken is shown in tables I and II. The majority of the data is not included in this report as it merely verifies the results.

The first important fact noted was that the efficiency of the burner as compared with the ideal value computed from Ref. (c), appeared to be excellent at high weight flows. See Figure 4. However, the burner would carbon up badly which indicated poor burning. This fact

was quickly verified by noting that there was only a very small flame in the vicinity of the fuel nozzle and that the temperature of the air six inches from the exit nozzle was less than 200° F which was entirely incompatible with the burner temperature reading. Further, a cool metal plate placed at the exit jet would condense out considerable fuel.

The only plausible explanation of the erroneous burner temperatures was that burning occurred on the thermocouple which would be probable under the circumstances. As no optical system was available to measure temperatures, all readings at high weight flow, where the flame was not in the vicinity of, or surrounding the thermocouple, were ignored in the final analysis.

This lack of combustion of the fuel at high weight flow was the result of too much cold air being blown at the flame. This cut down the vaporization of the fuel and, as pointed out in reference (c) by Jost, the ignition delay was greatly increased by the lowering of the mixture temperature. The two effects prevented the flame from progressing rapidly enough through the burner and the mixture was cooled below its' ignition temperature as it encountered more cool air. Reference (f) gives a cure for this situation. Mock would admit the primary air in better proportions by any one of his

suggested methods and the secondary air later after the mixture has burned almost completely.

Another method, impractical at the moment, would be to use a fast burning fuel such as hydrogen peroxide.

Another striking proof of the poor burning efficiency of this burner at high weight flow is to compare the curves on Fig. 10 where the exit nozzle areas are shown for .126 sq. in. and .385 sq. in. For an air-fuel ration of 80 it was found that the burner with the .126 sq. in. nozzle gave a greater temperature by 550 F. It must be realized that by increasing the exit area greater weight flows are obtained for the same air fuel ratio.

When equal weight flows were put through the burner, it is seen by Fig. 9. which was obtained from the effective part of Fig. 4. and from Fig. 8, that the efficiency of the combustion was only slightly increased for the .126 sq. in. nozzle over the .385 sq. in. nozzle. This is reasonable as within this range an increase in the burner pressure will increase the combustion efficiency slightly by increasing its speed of reaction. This is somewhat analogous to the advantage of a high compression internal combustion engine.

Examining the effect of varying fuel pressure, which in this case means varying fuel weight flow while

holding the exit nozzle area constant, it is apparent from Fig. 7 that combustion efficiency decreases with increased fuel flow. Fig. 7 is a combination of Fig. 5 and Fig. 6 and therefore shows the combustion efficiency for a total weight flow of .0502 lb. per sec. for various air-fuel ratios. In this particular case it indicates that at very low air-fuel ratios, near stoicheometric the burner efficiency again drops off. This result verifies the findings of References (d) and (f).

Fig. 10, for a constant air fuel ratio, shows that the smaller fuel pressure is more efficient. The explanation is the same as in the case of using the smaller exit nozzle area. In other words, the air flow must be larger with the higher fuel pressure to give the same air-fuel ratio, and this burner has already been proved to be better at the smaller weight flow.

Examining more closely the effect of air-fuel ratios near stoicheometric on efficiency, Fig. 10 shows clearly the loss in efficiency when one pictures the actual curves being dropped down to give the actual average temperature say approximately 300 F for the 40 lb. curve. This sudden drop in efficiency is the key to the rich blow out in the burner when using hydrocarbon fuels,

To review the blow out problem it will be recalled that with hydrocarbon fuels, kerosene and propane, the

blow out occurred at slightly above stoicheometric and that the flame was extinguished from the down stream end. In no case was the temperature ratio greater than 5.3. With hydrogen, blowout occurred starting with the up stream end and the temperature ratio was approximately 7. This immediately suggests that the hydrogen blow out was a flow effect while that of the hydrocarbon fuels were chemical and kinetic.

A reasonable explanation is that, as the hydrocarbon fuel is burned, many aldehydes and peroxides are formed in the flame front which require a great deal of oxygen while later, other unstable oxides such as H_2O_2 and CO_3 are formed requiring even more oxygen. As the oxygen available is limited, the mixture is heterogeneous and the time of combustion is small, the flame dies out for lack of oxygen necessarily form the down stream end. With hydrogen the only possible oxygen rich unstable products are H_2O_2 and HO_2 which may explain the difference.

It might be mentioned at this time that the presence of the aldehydes were painfully brought to the attention of the operator by their eye irritating qualities. It is regretted that time did not permit the running of rich blow out tests with oxygen enriched air and also with a homogeneous mixture of propane and air.

The above theory could also explain the poor

efficiencies at low air-fuel ratios. It was not a problem of insufficient oxygen to completely burn the fuel, but rather that the oxygen was tied up in heavy molecules or that the remaining free molecules did not have time to encounter the fuel at a burnable concentration while in the burner. This caused the flame to stand out through the nozzle and burn in the free air. A temperature traverse made in the free jet showed considerable rise in temperature in the shock diamonds, probably due to the conversion of some unstable carbon radical.*

In the beginning when the burner was being tested for proper length it was noted that instability was decreased for longer burner tubes. In other words that improving the combustion efficiency increased the air-fuel ratio at blow out. This fact has since been verified by reference (d).

Therefore, improving the combustion efficiency will not necessarily prevent blow out as the oxygen removing process will still continue. It is unlikely that this type of burner would be used at these low air-fuel ratios except for very short intervals of time. As a result the blow out problem at this point is the most important.

* This phenomenon was investigated by Lt.Comdr. C. J. Beers, U.S. Navy on the same burner.

During the rich blow tests a confirmation was made of the fact stated in reference (f) that liquid fuel is to be found on the up stream end of the burner liner despite the high temperatures involved. It is not surprising then that the burning of the flame was rough. It could easily happen, that, as this fuel flows down to a hotter part of the burner, it is vaporized with a flash back of the flame through the boundary layer. This would cause a subsequent pressure rise with less air supplied and then a rarefaction. During the rarefaction more fuel would be vaporized into an already rich mixture causing an even larger fluctuation. It was noted that the burning of propane was far smoother, and that of hydrogen smoother yet. This phenomenon should be further explored and the above explanation is only offered as a possible solution to the process whereby instability starts.

It was found during the runs with propane and hydrogen that not only stability was improved but that starting was simple and positive. This follows from the decreased specific gravity and increased volatility of the gaseous fuels. The difficulty in starting the burner with kerosene was due in part to the fact that it was not in the best location. However, the principle difficulties were caused by liquid fuel shorting out the plug and that

the plug carboned up after about ten minutes of running. While on the problem of starting, it should be stated that restarting the modern turbo jet at high altitudes where the static temperature is low is one of the serious drawbacks to the engine.

It might appear at this point that the use of a homogeneous air and fuel mixture would be easier to burn. However, it must be remembered that theflammability range of kerosene is approximately an air-fuel ratio of eight to twenty-seven. Experience has shown that even a homogeneous hydrocarbon mixture will not burn much below stoichiometric and therefore the range of flammability is too narrow. Precise control of primary and secondary air would overcome the difficulty, but even at best the problem of getting a true homogeneous mixture of a liquid fuel in the vicinity of the spark plug or pilot light is difficult.

Conclusions and Recommendations

As a result of the foregoing tests it can be stated that

(1) Increasing burner pressure will increase stability and lower the rich blow out limit. It will also improve combustion efficiency at all air-fuel ratios.

(2) The combustion efficiency of this burner is poor at high air-fuel ratios and that it could be improved by designing the burner to give better distribution of primary air.

(3) In this burner rich blow out may be caused by lack of free oxygen, and that probably the limit could be lowered by addition of oxygen or oxygen rich compounds.

(4) The greater weight flows through the burner, i.e. higher Mach numbers, give lower combustion efficiencies for the same air-fuel ratio.

(5) Longer burner tubes will improve combustion efficiency, but that they increase the rich blow out limit.

(6) Homogeneous mixtures are impractical for a burner which must operate under a wide range of air-fuel mixtures.

(7) It is very likely that liquid fuel in the burner liner is the cause of rough burning, or at least starts the instability.

The foregoing points with the exception of (2) and

(6) would make excellent quantitative studies. Combustion with flow can be called a new subject. It is extremely important at the present time.

The method of analysis presented is believed to be a quick and sure method of evaluating burners to meet given requirements, and especially to check modifications on the burner. The weak point, of course, is in temperature measurements, but this, unless it is done by optical means, is always a problem in combustion studies.

On the other hand gas analyses taken in this burner were not at all consistent with what the author feels was actually going on. Possibly the technique was poor, but the problem of the hot probe completing the reaction, liquid fuel on the liner and in the gas, plus the fact that a true picture could only be gotten from a three dimension traverse in velocity and gas analysis, make this method questionable, or at least time consuming.

Another method used in this report to check combustion efficiencies is quite accurate and quick. This consisted in using the weight flow equation

$$\frac{W\sqrt{T_o}}{PA} = M\sqrt{\frac{kg}{R}\left[1+\frac{(k-1)}{2}M_2\right]}$$

from reference (a) applied to the exit nozzle. If the pressure ratio was greater than critical M=1, P=atmospheric, W=known, A=A nozzle x nozzle coefficient and $k=\frac{cp}{cv}$ (Ref. h); R can be considered a constant with

little error. With these values To can be computed which gives an average temperature directly.

If the pressure ratio was less than critical, a total pressure and static pressure readings were taken to compute M.

The drawback in this method was that the readings were not taken in the burner and therefore heat transfer had taken place with resulting lowering of the temperature.

This report has given some of the factors which are important in combustion with flow. In a design of a burner, for any particular use they must be considered. It is not possible to design one simple burner for any situation any more than one could burn fuel oil in a gasoline engine.

Calculations

With the exception of weight flow all calculations are straight forward.

Air weight flow was computed as follows:

From reference (a)

$$\frac{Po}{P} = \left[1+\left(\frac{k-1}{2}\right) M^2\right]^{\frac{k}{k-1}}$$

Expanding in the binomial theorem

$$\frac{Po}{P} - 1 = \frac{k-1}{2} \times \frac{k}{k-1} M^2 + \ldots$$

$$\therefore \frac{Po-P}{P} \times \frac{2}{k} = M^2$$

$$M = \sqrt{\frac{2}{k} \quad \frac{\Delta P \text{ in Hg}}{P \text{ in Hg}}}$$

As $\frac{W\sqrt{To}}{PA} = f\ (M)$: correct slope of curve for low M

$$\therefore\ W = f\ (M) \times P \times A\sqrt{\frac{2}{k} \quad \frac{\Delta P}{P} \times \frac{1}{To}} = .2376 \sqrt{\frac{P \times \Delta P}{To}}$$

where f (M) = .4892 and A = .749 sq. in.; which has good accuracy up to ΔP = 3.5" of Hg.

Table I

Data and Computations of Combustion Runs

Area of Nozzle = .26 sq. in. Barometer = 30.05

All Pressures in (in. Hg.) All Temp. in (deg.R)

Run No.	P (fuel)	P_1 (Abs)	ΔP (1-2)	ΔP (2-3)	ΔP (2-4)	P_2 (Abs)	P_4 (Abs)	T_1	T_2
1.a	40	89.25	.7	.4	.3	88.55	88.25	538	988
b	40	81.35	.55	.3	.3	80.80	80.50	541	1105
c	40	71.75	.45	.3	.25	71.30	71.05	542	1525
d	40	60.55	.25	.2	.20	60.30	60.10	544	2132
e	40	40.05	.05	.05	.05	40.00	39.95	544	2863
2.a	60	90.15	.45	.3	.40	89.70	89.30	548	1700
b	60	80.85	.35	.3	.35	80.50	80.15	549	2168
c	60	70.35	.225	.2	.25	70.12	69.87	550	2509
d	60	60.55	.20	.15	.20	60.35	60.15	550	2836
e	60	40.05	.10	.10	.10	39.95	39.85	551	2895
3.a	80	90.55	.35	.30	.30	90.20	89.90	552	2213
b	80	80.55	.35	.25	.30	80.20	79.90	552	2527
c	80	70.15	.225	.20	.20	69.93	69.73	552	2852

Table I
(Continued)

Data and Computations of Combustion Runs

Area of Nozzle = .126 sq. in. Barometer = 30.05

All Pressures in (in. Hg.) All Temp. in (deg. R)

Run No.	W (fuel)	W (air)	W (total)	Air/Fuel	T_2/T_1	Comments
1.a	.00045	.0772	.07765	175	1.84	
b	.00046	.0677	.06816	147	2.04	
c	.00047	.0578	.05827	123	2.81	Very Stable
d	.00048	.0404	.04088	84	3.90	
e	.00050	.0144	.01490	29	5.27	Unstable
2.a	.00108	.0644	.06548	60	3.10	
b	.00125	.0539	.05515	43	3.95	Very Stable
c	.00130	.0405	.04180	31	4.56	
d	.00136	.0351	.03646	25.8	5.15	
e	.00140	.0202	.02162	15	5.26	Unstable
3.a	.00136	.0569	.05826	42	4.00	Very Stable
b	.00150	.0535	.05500	35.7	4.57	
c	.00170	.0400	.04170	23.5	5.17	Unstable

Table II

Data and Computations of Combustion Runs

Area of Nozzle = .385 sq. in. Barometer = 30.05

All Pressures in (in. Hg.) All Temp. in (deg R)

Run No.	P (fuel)	P_1 (Abs)	P (1-2)	P (2-4)	P_2 (Abs)	P_4 (Abs)	T_1	T_2
1.a	100	89.95	5.25	1.95	84.70	82.75	538	1331
b	100	81.85	4.25	1.60	77.60	76.00	537	1437
c	100	70.55	3.10	1.30	67.45	67.15	538	1618
d	100	60.95	2.10	.90	58.85	57.95	539	1858
e	100	55.45	1.60	.80	53.85	53.05	541	2024
f	100	51.35	1.40	.90	49.95	49.05	543	2329
2.a	80	50.55	1.30	.80	49.25	48.45	543	2250
b	80	40.45	.50	.50	39.95	39.45	544	2860
c	80	60.15	2.15	.95	58.00	57.05	545	1880
d	80	70.25	3.10	1.25	67.15	65.90	547	1657
e	80	81.05	4.10	1.60	76.95	75.35	546	1492
f	80	91.25	5.10	1.90	86.15	84.25	547	1340
3.a	40	87.35	6.75	2.25	80.60	78.35	547	953
b	40	83.85	5.85	2.05	78.00	75.95	549	968
c	40	70.25	4.10	1.50	66.15	64.65	550	1183
d	40	61.75	3.10	1.10	58.65	58.55	550	1323
e	40	50.55	1.75	.80	48.80	48.00	551	1682
f	40	42.05	.90	.55	41.15	40.60	552	2132

Table II

Data and Computations of Combustion Runs

Area of Nozzle = .385 sq. in. Barometer = 30.05

All Pressures in (in. Hg.) All Temp, in (deg R)

Run No.	W (fuel)	W (air)	W (total)	Air/Fuel	T_5/T_1	Comments
1. a	.00195	.2160	.21795	112	2.47	
b	.00195	.1865	.18845	95.6	2.63	
c	.00205	.1481	.15015	72	3.00	
d	.00205	.1136	.11565	55.5	3.44	
e	.00208	.0946	.09668	45.5	3.74	
f	.00210	.0845	.0866	41.6	4.28	Unstable
2. a	.00202	.0815	.08352	39	4.15	
b	.00208	.0455	.04763	21.8	5.27	Unstable
c	.00208	.1161	.11818	56	3.45	
d	.00219	.1466	.14879	67	3.02	
e	.00195	.1805	.18245	92.5	2.73	
f	.00150	.2125	.21400	123	2.45	
3. a	.00104	.2370	.23804	228	1.74	
b	.00098	.2165	.21748	221	1.76	
c	.00110	.1669	.16800	151	2.15	
d	.00114	.1368	.13794	120	2.41	
e	.00125	.0935	.09475	74.5	3.05	
f	.00164	.0648	.06644	39.5	3.76	Unstable

Section C

Curves

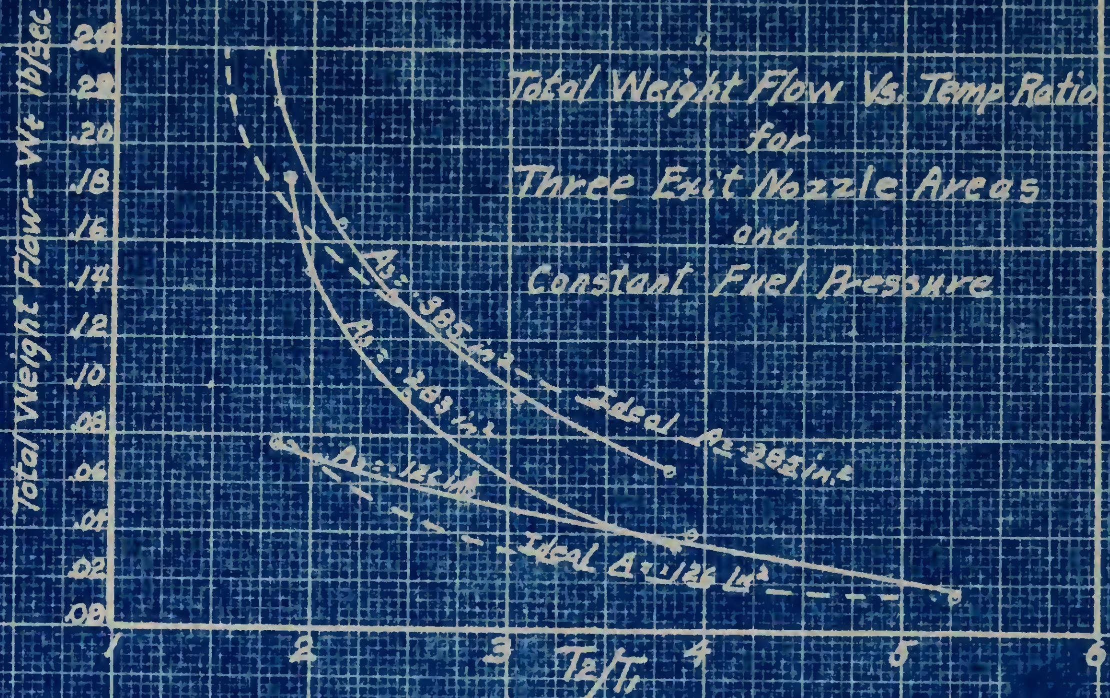

Total Weight Flow Vs. Temp Ratio
for
Three Exit Nozzle Areas
and
Constant Fuel Pressure
Total Weight Flow - Wt lb/sec
T2/T1

Fig. 4

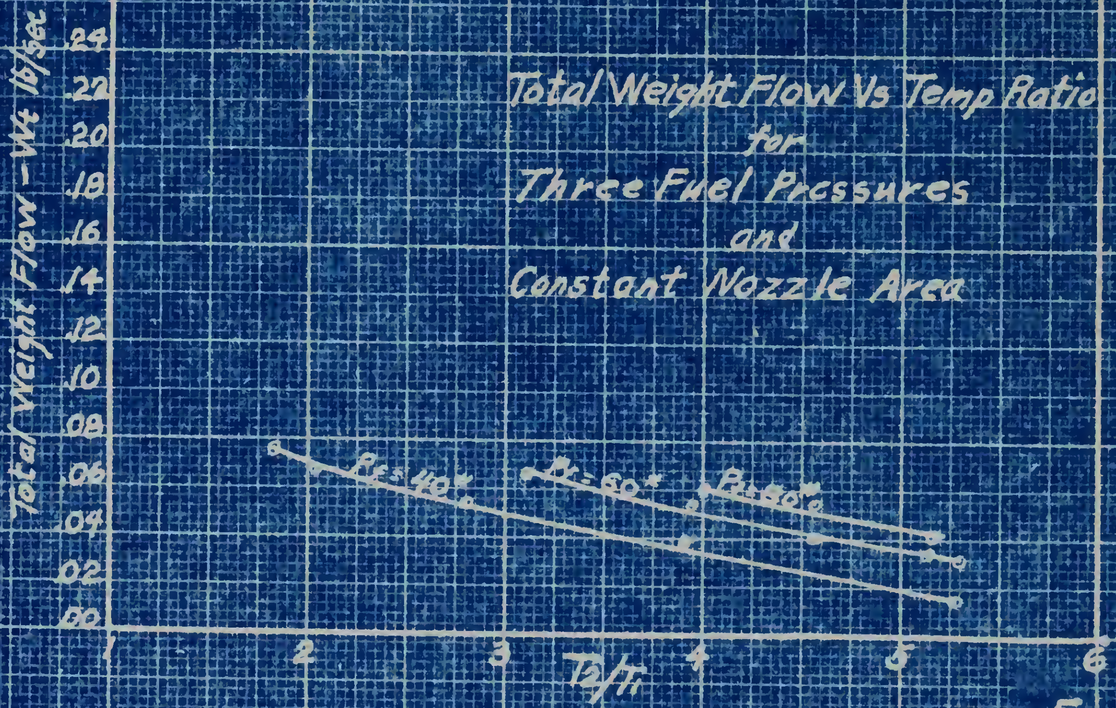

Total Weight Flow Vs Temp Ratio
for
Three Fuel Pressures
and
Constant Nozzle Area
Total Weight Flow - Wt lb/sec
T2/T1

Fig. 5

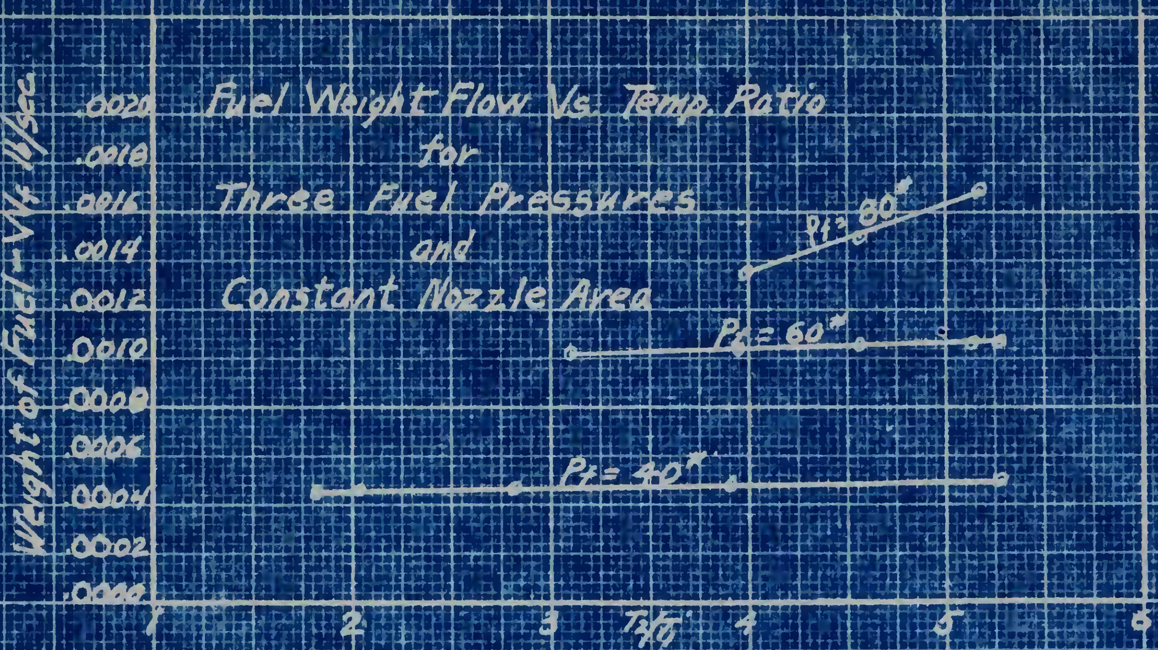

Fuel Weight Flow Vs. Temp. Ratio
for
Three Fuel Pressures
and
Constant Nozzle Area
Weight of Fuel - Wf lb/sec
.0020
.0018
.0016
.0014
.0012
.0010
.0008
.0006
.0004
.0002
.0000
Pf = 80#
Pf = 60#
Pf = 40#
1
2
3
T2/T1
4
5
6

Fig. 6

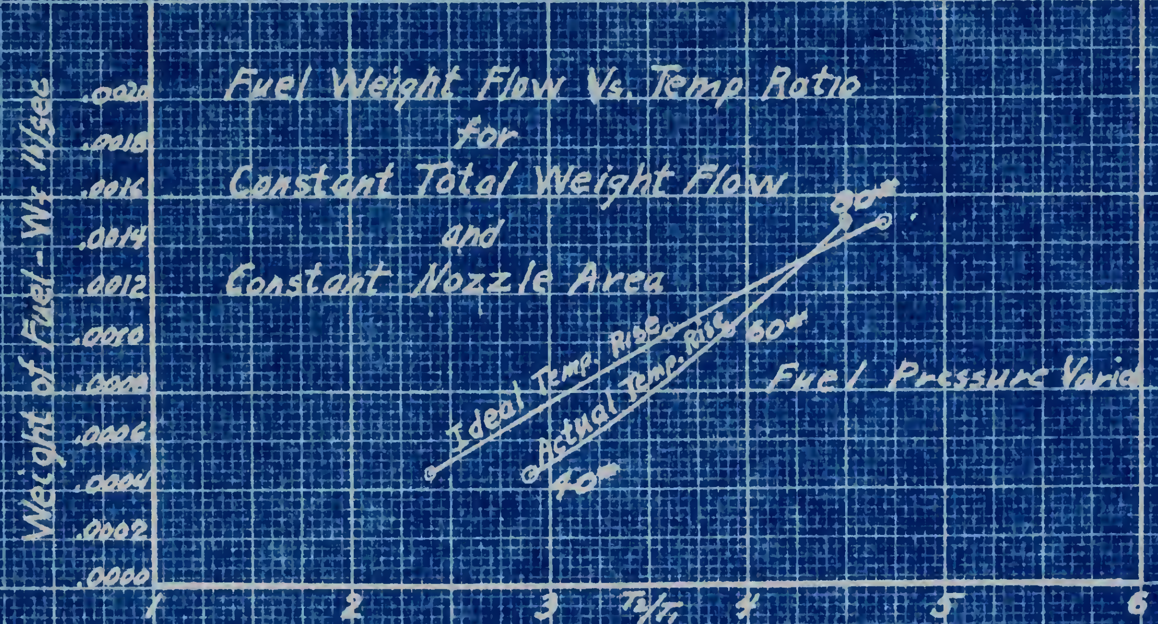

Fuel Weight Flow Vs. Temp Ratio
for
Constant Total Weight Flow
and
Constant Nozzle Area
Weight of Fuel - Wf lb/sec
.0020
.0018
.0016
.0014
.0012
.0010
.0008
.0006
.0004
.0002
.0000
80#
60#
40#
Ideal Temp Rise
Actual Temp Rise
Fuel Pressure Varied
1
2
3
T2/T1
4
5
6

Fig 7

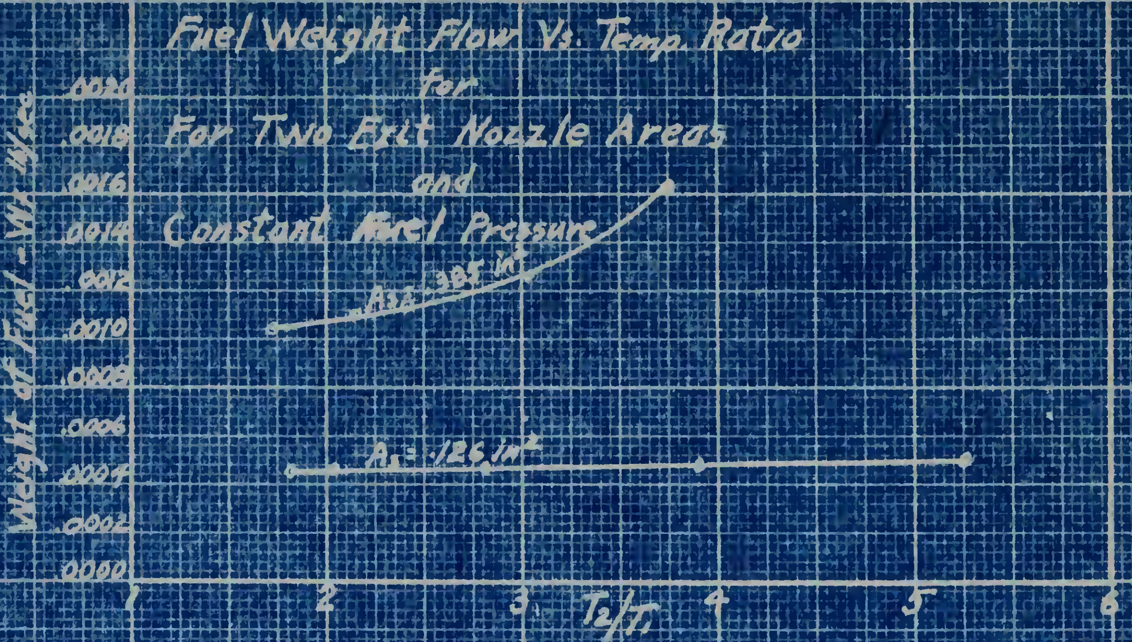

Fuel Weight Flow Vs. Temp Ratio
for
For Two Exit Nozzle Areas
and
Constant Fuel Pressure
Weight of Fuel - Wf lb/sec
$A_3 = .385\ in^2$
$A_3 = .125\ in^2$
T_2/T_1

Fig 8

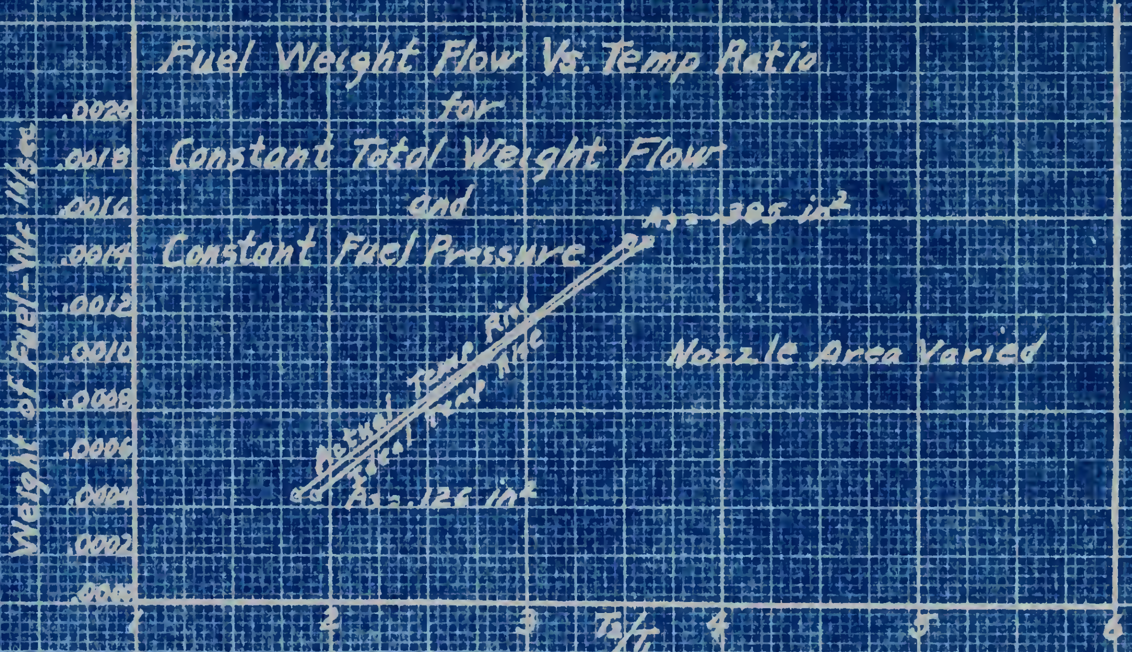
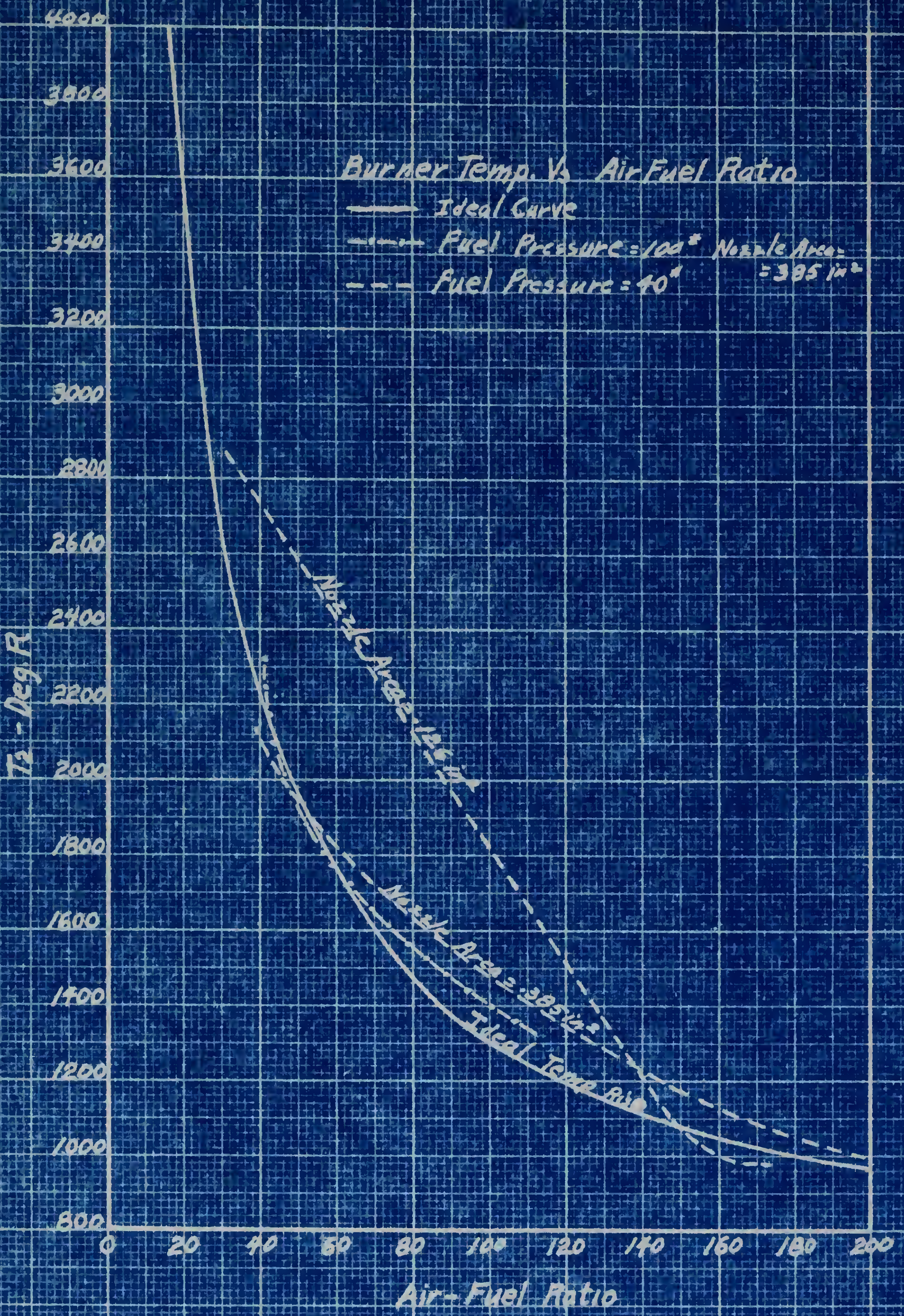

Fuel Weight Flow Vs. Temp Ratio
for
Constant Total Weight Flow
and
Constant Fuel Pressure
Weight of Fuel - Wf lb/sec
$A_3 = .385\ in^2$
$A_3 = .125\ in^2$
Actual Temp Rise
Ideal Temp Rise
Nozzle Area Varied
T_2/T_1

Fig. 9



Burner Temp. Vs Air Fuel Ratio
Ideal Curve
Fuel Pressure = 100# Nozzle Area = .385 in²
Fuel Pressure = 40#
T2 - Deg R
4000
3800
3600
3400
3200
3000
2800
2600
2400
2200
2000
1800
1600
1400
1200
1000
800
0
20
40
60
80
100
120
140
160
180
200
Air-Fuel Ratio
Nozzle Area
Nozzle Area = .385 in²
Ideal Temp

Fig. 10

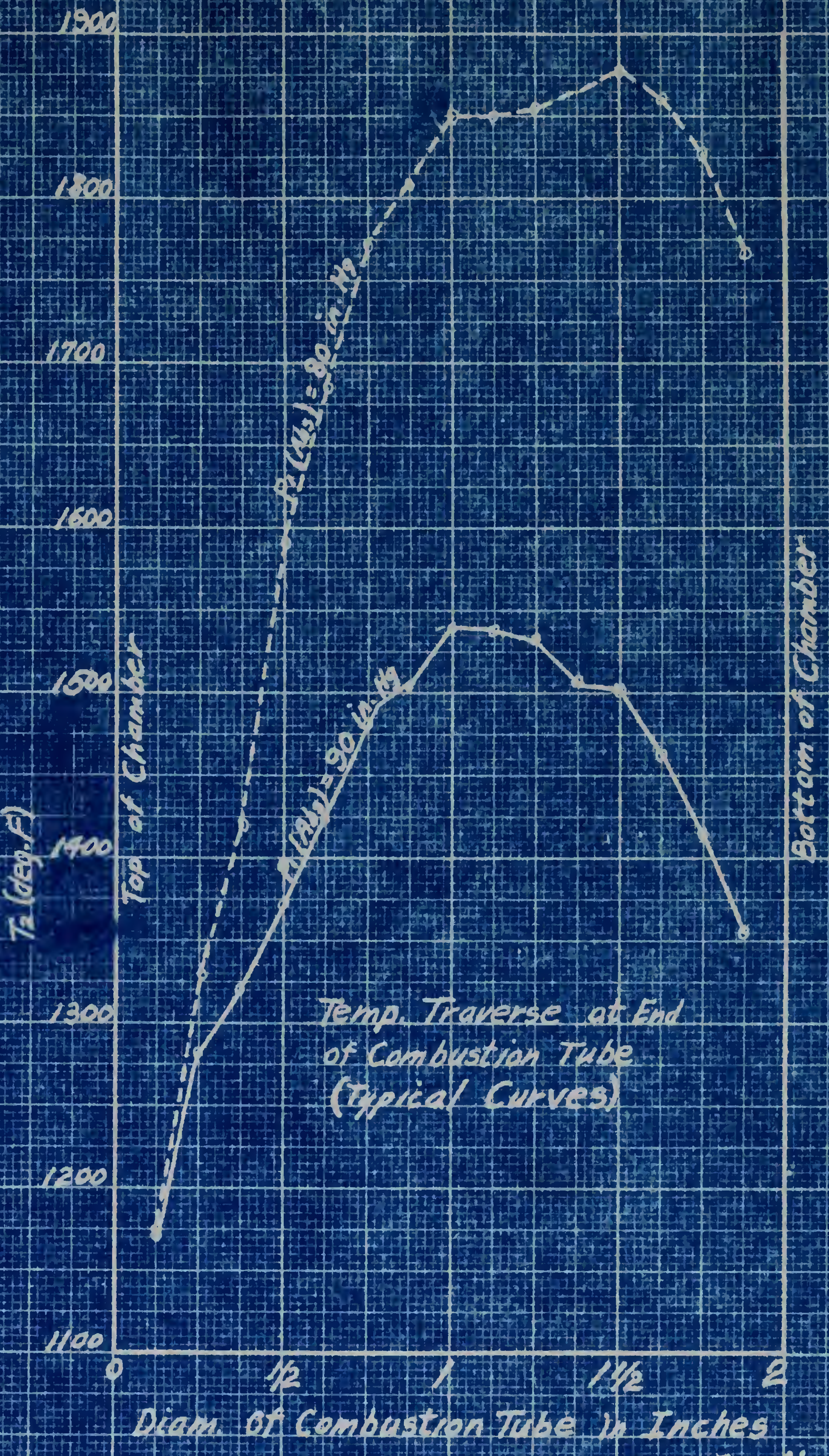

1900
1800
1700
1600
1500
1400
1300
1200
1100
T_2 (deg. F)
Top of Chamber
Bottom of Chamber
P_2(Abs) = 80 in. Hg
P_2(Abs) = 90 in. Hg
Temp. Traverse at End
of Combustion Tube
(Typical Curves)
0
1/2
1
1 1/2
2
Diam. of Combustion Tube in Inches

Fig. 11

Section D

Photographs

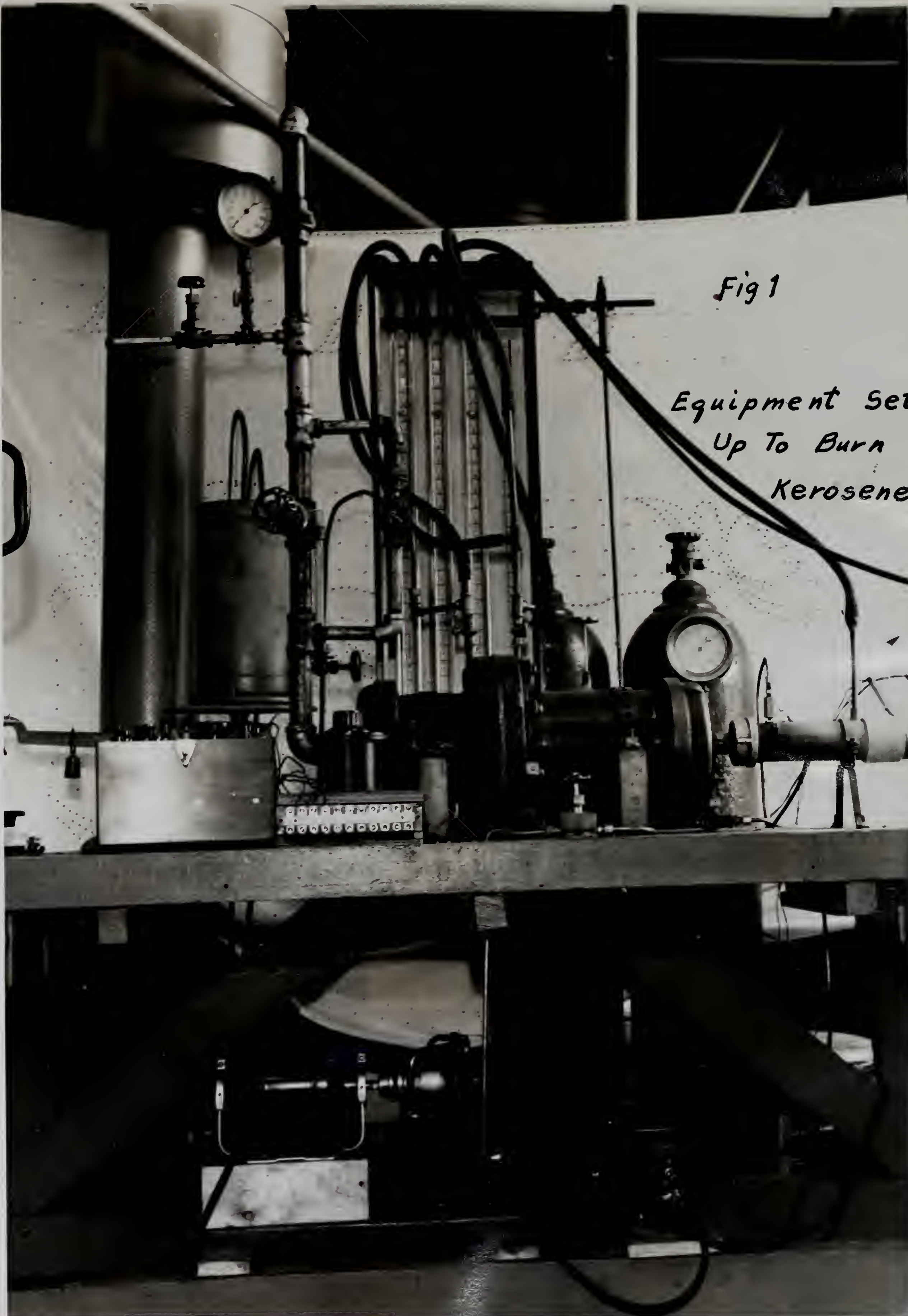


Fig 1

Equipment Set Up To Burn Kerosene

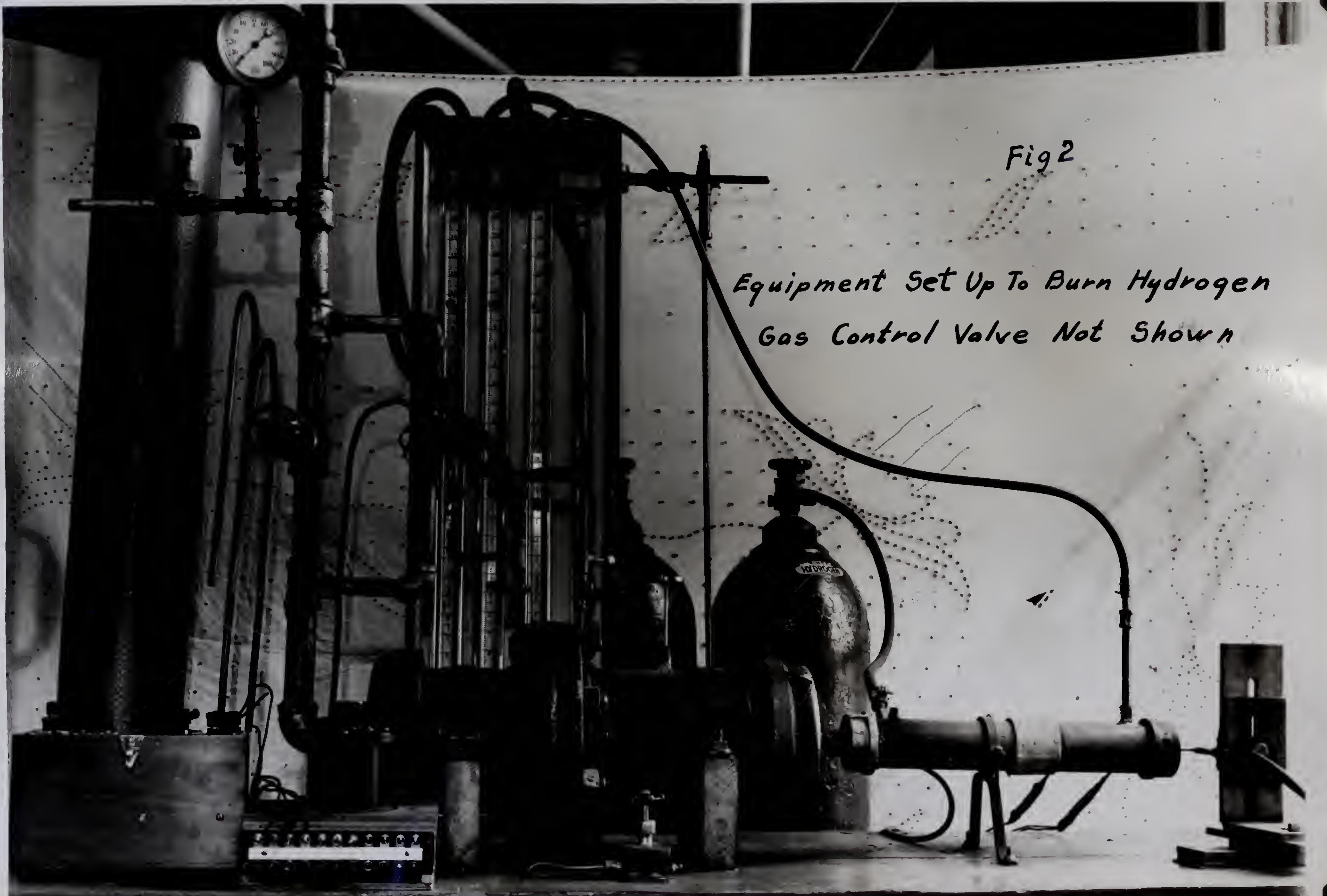
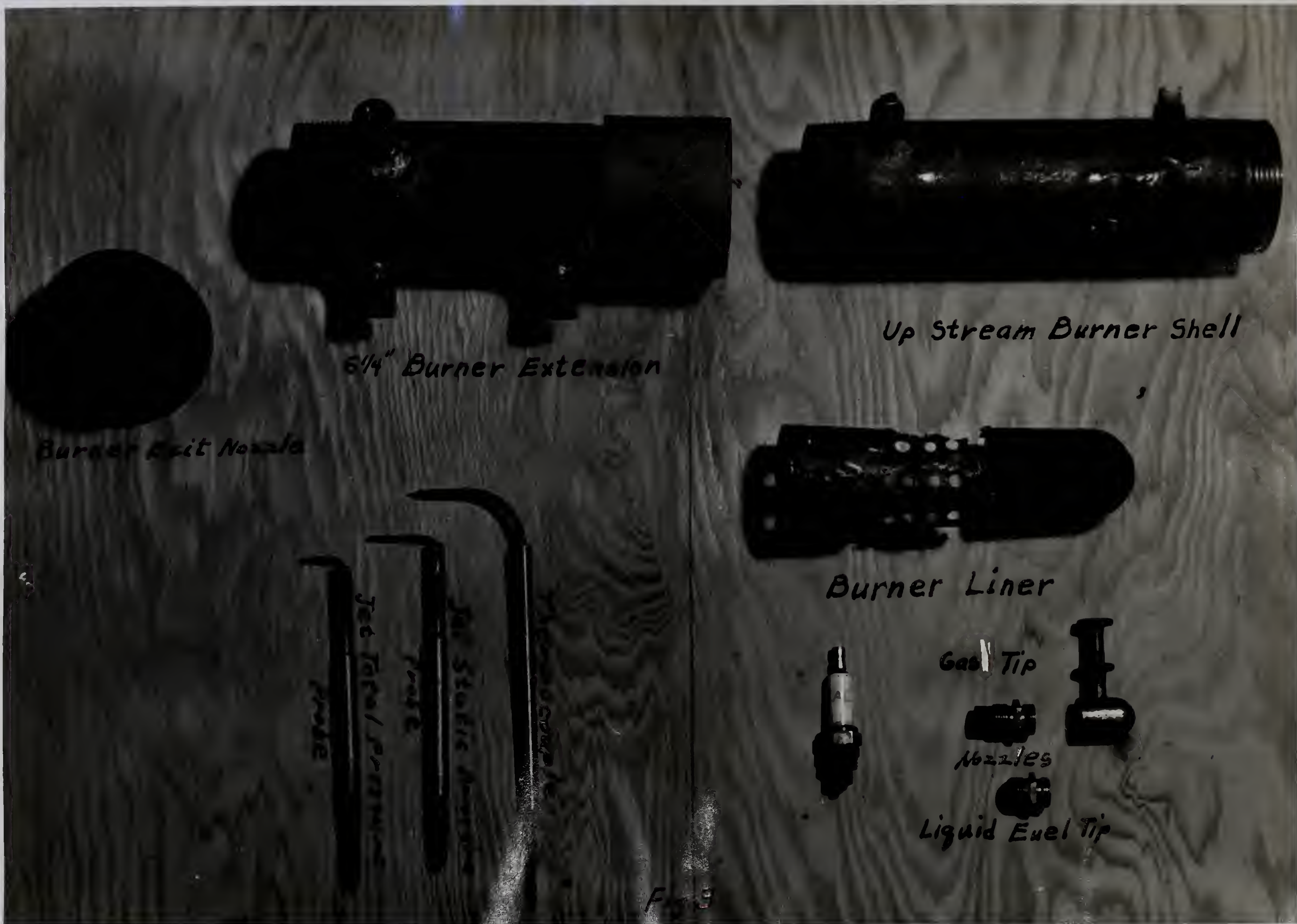


Fig 2

Equipment Set Up To Burn Hydrogen
Gas Control Valve Not Shown


6¼" Burner Extension
Up Stream Burner Shell
Burner Liner
Gas Tip
Nozzles
Liquid Fuel Tip

Symbols and Nomenclature.

Symbol	Quantity	Units
A	Area	in^2
C_p	Specific heat at constant pressure	B.T.U./lb°F
C_v	Specific heat at constant volume	B.T.U./lb°F
G	Standard acceleration of gravity	$ft./sec^2$
k	Specific heat ratio $\frac{C_p}{C_v}$	none
M	Mach number	none
P	Absolute static pressure	lb/in^2
Po	Absolute total pressure	lb/in^2
R	Gas constant	ft.lb/lb°F
T	Absolute static temperature	°Rankine
To	Absolute total temperature	°Rankine
V	Velocity	ft./sec.
W	Weight flow per unit time	lb./sec.
Wa	Weight of air flowing per unit time	lb./sec.
Wf	Weight of fuel flowing per unit time	lb./sec.
ρ	Density	$slugs/ft^3$
w	Weight flow	

References.

a. The Thermodynamics of Air at High Velocities: Neil P. Bailey, Journal of the Aeronautical Sciences, July 1944, Volume 11, No. 3.

b. Gas Turbine Combustion and Stability: Neil P. Bailey. Mechanical Engineering Department, Rensselaer Polytechnic Institute, Troy, N. Y., Sept. 1947

c. Explosion and Combustion Processes in Gases: Jost Wilhelm. 1946, 1st ed. McGraw-Hill Book Co. Inc., New York and London.

d. Combustion in Moving Air: Frank R. Caldwell, Fillmer W. Ruegg, and Lieff O. Olsen, S.A.E. preprint 181 of April 1948.

e. Ideal Temperature Rise From Constant Pressure Combustion of Hydrocarbon Fuels: Newman A. Hall, Richard C. Mulready; S.A.E. preprint of April 1948

f. Engineering Development of the Jet Engine and Gas Turbine Burner: Frank C. Mock S.A.E. preprint of January 1946.

g. Combustion, Flames and Explosions of Gasses; Lewis and von Elbe. University Press, 1938

h. Mean Specific Heats for Working Media of Gas Turbine Power Plants, Newman A. Hall, S.A.E. Quarterly Transactions, Vol. 1, 490-497.

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B175 Balch
c.1 Stability with combustion and flow.